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# RESEARCH MEMORANDUM

SOME OBSERVATIONS OF SHOCK-INDUCED TURBULENT  
SEPARATION ON SUPERSONIC DIFFUSERS

By T. J. Nussdorfer

Lewis Flight Propulsion Laboratory  
Cleveland, Ohio

FOR REFERENCE

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## NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

RESEARCH MEMORANDUMSOME OBSERVATIONS OF SHOCK-INDUCED TURBULENT SEPARATION  
ON SUPERSONIC DIFFUSERS

By T. J. Nussdorfer


## SUMMARY

A survey of experimental data at supersonic speed indicates that shock-induced separation of a turbulent boundary layer will result for Mach numbers of approximately 1.33 or greater when a theoretical stream static-pressure-rise ratio of approximately 1.89 occurs across a shock interacting with the boundary layer. The significance of this tentative criterion for turbulent-boundary-layer separation is discussed with respect to the design of supersonic diffusers.

## INTRODUCTION

In supersonic flow, shock waves often create adverse pressure gradients far in excess of those encountered in subsonic flow. Whenever an adverse pressure gradient exists in the presence of a boundary layer, some of the boundary-layer air may have insufficient momentum to penetrate the higher pressure region (even with mixing), and thus a reverse flow and a region of boundary-layer separation may develop along the surface. One cause of separation is a strong shock interacting with a boundary layer. Therefore, it may be possible to deduce an empirical criterion for boundary separation from a study of shocks interacting with the boundary layer.

Three basic types of shock interacting with the boundary layer have been discussed in references 1 and 2 and are shown in figure 1. The normal shock (fig. 1(a)) occurs at Mach numbers below approximately 1.3 and is usually straight and normal to the flow. No separation appears to follow a normal shock. The curved shock (fig. 1(b)) changes inclination continuously with increasing distance from the wall. Separation usually occurs behind a curved shock, but there is also a strong tendency toward reattachment. The branched shock (fig. 1(c)) starts out as an oblique shock and is characterized by a discontinuous change in angle at some distance from the surface. Separation behind a branched shock is extensive in nature and shows little tendency to reattach. Thus the existence of the branched-shock pattern at the boundary layer may be used as an indication of shock-induced separation.



The results of theoretical analyses of the branched-shock phenomena are presented in references 1 to 3. In these analyses the effect of the boundary layer was neglected. By assuming the sum of the deflections of streamlines through shocks a and b of figure 1(c) to be equal to the deflection through shock c, and by assuming the product of the pressure ratios across a and b to be equal to the pressure ratio across c, Weise (ref. 3) related the shock configuration to the Mach number. For any given Mach number above 1.24, branched shocks are possible; the exact configuration is dependent upon any one of the deflections a, b, or c in the system. The action of the boundary layer is apparently the determining factor in the orientation of the branched shock and concomitant separation.

Experimental reports on linear expansion nozzles (ref. 4 and unavailable reports) indicated that when a boundary layer was present the branched shock occurred for Mach numbers greater than about 1.35 to 1.4, depending upon the nozzle expansion angle. For Mach numbers less than these, a normal shock without separation was observed. Therefore, it appears that the existence of boundary-layer separation is dependent upon the stream static-pressure-rise ratio.

The work reported in reference 4 is for turbulent boundary layers. From the results of reference 5, a marked difference in the type of separation and point of separation should be expected between turbulent and laminar boundary layers. Inasmuch as turbulent mixing is much more effective than molecular mixing in transferring momentum within a boundary layer, separation would be expected for a laminar boundary layer for smaller values of pressure rise than that required for a turbulent boundary layer. Extension of Gruschwitz calculations to cover separation in transonic flow with shocks is included in reference 6. A more complete discussion of separation is given in reference 7.

In the absence of a theoretical explanation of shock-induced separation of a turbulent boundary layer, an engineering criterion obtained from a survey of experimental data has been deduced. This report presents the tentative criterion, which relates separation or nonseparation of the boundary layer to the theoretical static-pressure-rise ratio across an imposed shock. The significance of the criterion is discussed with regard to supersonic diffusers for ram-jet and turbojet engine application.

The criterion presented in this report was developed at the NACA Lewis laboratory in 1951, but publication was withheld at that time because of parallel studies presented in reference 8. The information contained in reference 8 has since been superseded by reference 9. The recent work of reference 10, which includes different criteria for predicting shock-induced boundary-layer separation from those of reference 9, supports the conclusions presented herein. Release of this paper in substantially the original form is, therefore, considered appropriate.

## DISCUSSION

In this report separation was distinguished by the presence of a branched shock. Separation was most easily recognized from a schlieren or interferometer photograph, but velocity and total-pressure profiles and static pressures in the region of the boundary layer were also useful. Most of the data presented (refs. 11 to 14) were obtained from studies on supersonic diffuser inlets. Investigation of these inlets over a range of stream Mach numbers provides a convenient method of studying the interaction of shocks of varying strength upon the boundary layer. The first inlets studied were of the two-dimensional ramp type where the angle  $\lambda$  which the ramp makes with the free stream adequately describes the inlet for this study. For a given free-stream Mach number, a theoretical static-pressure-rise ratio across the normal shock may be obtained for any given ramp angle. The theoretical curves of figure 2 relate the ramp angle and the free-stream Mach number to various values of static-pressure rise across the normal shock. In the Mach number range from 1.0 to 2.0, a value of theoretical static-pressure-rise ratio of approximately 1.89 appears to define the regions of separation and no separation on the basis of the data presented in figure 2.

A plot similar to figure 2 was made for a conical three-dimensional diffuser inlet (see fig. 3). In this case, the static-pressure-rise ratio is based on the theoretical nonviscous cone surface Mach number. Again from the data of figure 3, a theoretical static-pressure-rise ratio across the normal shock of 1.89 appears to define the separation and nonseparation regions in the Mach number range from 1 to 3.

When the normal shock occurs at Mach numbers less than approximately 1.33, a curved shock instead of a normal (see fig. 1) was sometimes observed to interact with the boundary layer, particularly for low values of Reynolds number. An example of the curved shock changing to a branched shock is shown in the schlieren photographs (fig. 4) of points A, B, and C of figure 2. Most of the data shown in figure 2 were obtained during an investigation of side inlets which has been reported in part in reference 14. The Reynolds number in the region of the normal shock for these inlets is 900,000 based upon the distance from the leading edge to the shock. It should be noted that the inlet ramp was located immediately adjacent to the turbulent boundary layer of the body (Reynolds number, 29,000,000). It is therefore very likely that transition has been forced on the ramp inlet by the outer extremities of the body boundary layer, even though the Reynolds number at the normal shock is only 900,000.

That the value of static-pressure-rise ratio appears to be a useful correlating parameter for separation is better illustrated in figure 5. The data from figures 2 and 3 are replotted to show the variation of Mach number ahead of a normal shock and the theoretical static-pressure-rise ratio across that shock. The data presented cover a Reynolds number

range from 900,000 to 5,000,000 based on the length of wetted surface ahead of the shock. It is clear from the figure that occurrence of a normal shock at a local Mach number of 1.33 or greater is a good engineering rule for the prediction of the occurrence of separation on the compression surface of two- and three-dimensional supersonic inlets.

In many experimental studies of supersonic inlets (refs. 13 and 15, e.g.), separation of the flow on conical centerbodies has been indicated as a source of instability in the inlet. The presence of a separated boundary layer has also been found to be the cause of large losses in pressure in the subsonic diffuser. If the supersonic inlet can be designed, therefore, so as to provide a supersonic region of Mach number less than 1.33 in which to position the normal shock, no separation should occur in this region. A similar empirical criterion for the design of conical supersonic diffusers to avoid separation (ref. 16) recommends that the cone surface Mach number at the inlet not exceed 1.3.

In the case of inlets having one oblique shock and one normal shock with no internal compression, a large enough compression can be obtained across the oblique shock to limit the Mach number behind it to a value less than 1.33. The effect of this limitation on the theoretical pressure recovery obtained by neglecting the subsonic losses is shown for ramp-type two-dimensional inlets and conical three-dimensional inlets at various free-stream Mach numbers in figure 6. In the Mach number range from 1.5 to 2.0, the inlets may be designed for near-maximum pressure recoveries and still not encounter boundary-layer separation. Above a Mach number of approximately 2, the initial compression required to avoid separation is larger than the optimum for pressure recovery.

It is to be expected that a shock of a given strength interacting with a turbulent boundary layer would have the same effect whether it be induced by a blunt body or a supersonic diffuser inlet. Results of several blunt-body investigations are presented in references 17 to 19. The shock angles were computed from static-pressure measurements and did not agree with the measured shock angles. Inasmuch as the actual shock angles are the basis for the separation criterion, the measured shock angles obtained from the data reported in references 17 to 19 were used to compute the theoretical static-pressure-rise ratio reported herein (see fig. 5). These data show remarkable agreement with the value of 1.89 determined from the supersonic-diffuser data. It appears, therefore, that while the measured static-pressure-rise ratio in the boundary layer required for shock-induced separation may vary, the presence of separation is evidenced by a theoretical static-pressure-rise ratio of 1.89 across the shock.

Another simple criterion for shock-induced turbulent separation has been suggested by Nitzberg and Crandall (ref. 20). From a survey of many calculations applying the Gruschwitz method to subsonic airfoils

they found that as a first approximation the relation  $\left(\frac{u_{\text{separation}}}{u_{\text{initial}}}\right)^2 = \frac{1}{2}$  was valid. They suggest that this relation might also apply across shocks, giving  $\left(\frac{u_{\text{behind shock}}}{u_{\text{ahead of shock}}}\right)^2 = \frac{1}{2}$ . This relation, when rewritten in terms of static-pressure-rise ratio, gives essentially 1.89, the criterion of this report. Additional airfoil data reported in reference 18 indicated that the shock-stall Mach number (stream Mach number at which a large degree of boundary-layer separation occurs) was observed when local Mach numbers on the wing approached 1.33.

For convenience in predicting shock-induced separation for the case of oblique shocks, theoretical flow deflections and shock angles in two-dimensional flow that give a static-pressure-rise ratio of 1.89 are shown as a function of Mach number in figure 7. The amount of deflection possible without causing separation increases rapidly with increasing Mach number and reaches a maximum of  $13^\circ$  at a Mach number of 1.8. Above a Mach number of 1.8, a gradual decrease in the permissible flow deflection angle to avoid separation occurs.

#### SUMMARY OF RESULTS

From a survey of experimental supersonic flow data reported from varied sources, the following results were obtained:

1. Shock-induced separation of a turbulent boundary layer resulted for Mach numbers of approximately 1.33 or greater when a theoretical stream static-pressure-rise ratio of approximately 1.89 occurred across a shock interacting with the boundary layer.

2. Single oblique-shock supersonic inlets designed to prevent flow separation at the intersection of the normal shock and the boundary layer could obtain approximately the maximum pressure recovery for free-stream Mach numbers up to approximately 2.0. Above a Mach number of 2.0, optimum pressure recoveries will not be obtained if separation is to be avoided.

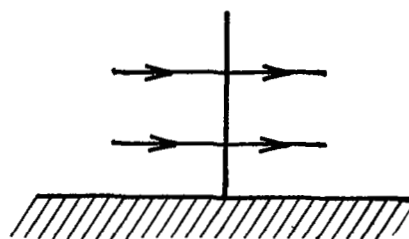
Lewis Flight Propulsion Laboratory  
National Advisory Committee for Aeronautics  
Cleveland, Ohio, January 24, 1954

## REFERENCES

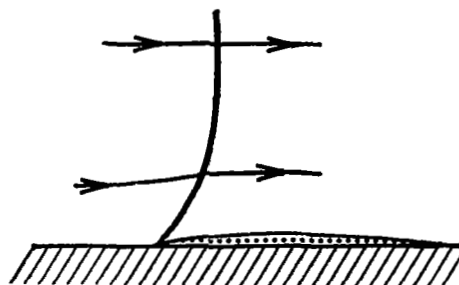
1. Eggink, H.: Compression Shocks of Detached Flow. NACA TM 1150, 1947.
2. Weise, A.: The Separation of Flow Due to Compressibility Shock. NACA TM 1152, 1947.
3. Weise, A.: Theory of the Branched Shock Wave. Tech. Trans. TT-16, National Research Labs. (Ottawa), June 4, 1947.
4. Eggink: Flow Development and Pressure Recovery in Supersonic Wind-Tunnels. L.F.A. Volkenrode, Trans. 52, British M.O.S.
5. Liepmann, H. W., Roshko, A., and Dhawan, S.: On Reflection of Shock Waves from Boundary Layers. NACA Rep. 1100, 1952. (Supersedes NACA TN 2334.)
6. Nitzberg, Gerald E., and Crandall, Stewart: Some Fundamental Similarities Between Boundary-Layer Flow at Transonic and Low Speeds. NACA TN 1623, 1948.
7. Schubauer, G. B., and Klebanoff, P. S.: Investigation of Separation of the Turbulent Boundary Layer. NACA Rep. 1030, 1951. (Supersedes NACA TN 2133.)
8. Donaldson, Coleman duP., and Lange, Roy H.: Study of the Pressure Rise Across Shock Waves Required to Separate Laminar and Turbulent Boundary Layers. NACA TN 2770, 1952. (Supersedes NACA RM L52C21.)
9. Lange, Roy H.: Present Status of Information Relative to the Prediction of Shock-Induced Boundary-Layer Separation. NACA TN 3065, 1954.
10. Bogdonoff, S. M., Kepler, C. E., and Sanlorenzo, C. E.: A Study of Shock Wave Turbulent Boundary Layer Interaction at  $M = 3$ . Rep. No. 222, Dept. Aero. Eng., Princeton Univ., July 1953.
11. Davis, Don D., Jr., and Wood, George P.: Preliminary Investigation of Reflections of Oblique Waves from a Porous Wall. NACA RM L50G19a, 1950.
12. Nussdorfer, Theodore J., Obery, Leonard J., and Englert, Gerald W.: Pressure Recovery, Drag, and Subcritical Stability Characteristics of Three Conical Supersonic Diffusers at Stream Mach Numbers from 1.7 to 2.0. NACA RM E51H27, 1952.

- 2458
13. Ferri, Antonio, and Nucci, Louis M.: The Origin of Aerodynamic Instability of Supersonic Inlets at Subcritical Conditions. NACA RM L5OK30, 1951.
  14. Schueller, Carl F., and Esenwein, Fred T.: Analytic and Experimental Investigation of Inlet-Engine Matching for Turbojet-Powered Aircraft at Mach Numbers up to 2.0. NACA RM E51K20, 1952.
  15. Dailey, C. L., McFarland, H. W., and Scobee, R.: Development of Ramjet Components. Prog. Rep. 9961-12, Aug., Sept., and Oct. 1950, Aero. Lab., Univ. Southern Cal., Nov. 7, 1950. (Navy Contract Noa(s) 9961.)
  16. Dailey, C. L.: Diffuser Instability in Subcritical Operation. Univ. Southern Cal., Sept. 26, 1950.
  17. Moeckel, W. E.: Flow Separation Ahead of Blunt Bodies at Supersonic Speeds. NACA TN 2418, 1951.
  18. Moeckel, W. E.: Flow Separation Ahead of a Blunt Axially Symmetric Body at Mach numbers 1.76 to 2.10. NACA RM E51L25, 1951.
  19. Moeckel, W. E.; and Evans, P. J., Jr.: Preliminary Investigation of Use of Conical Flow Separation for Efficient Supersonic Diffusion. NACA RM E51J08, 1951.
  20. Nitzberg, Gerald E., and Crandall, Stewart: A Study of Flow Changes Associated with Airfoil Section Drag Rise At Supercritical Speeds. NACA TN 1813, 1949.

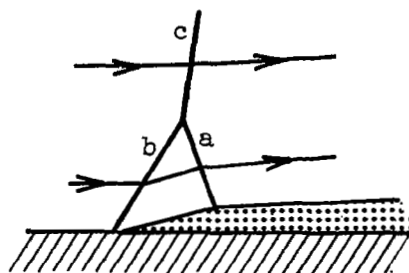




(a) Normal shock.



(b) Curved shock.



(c) Branched shock.



Figure 1. - Types of shock interacting with boundary layer in supersonic flow (refs. 1 and 2).

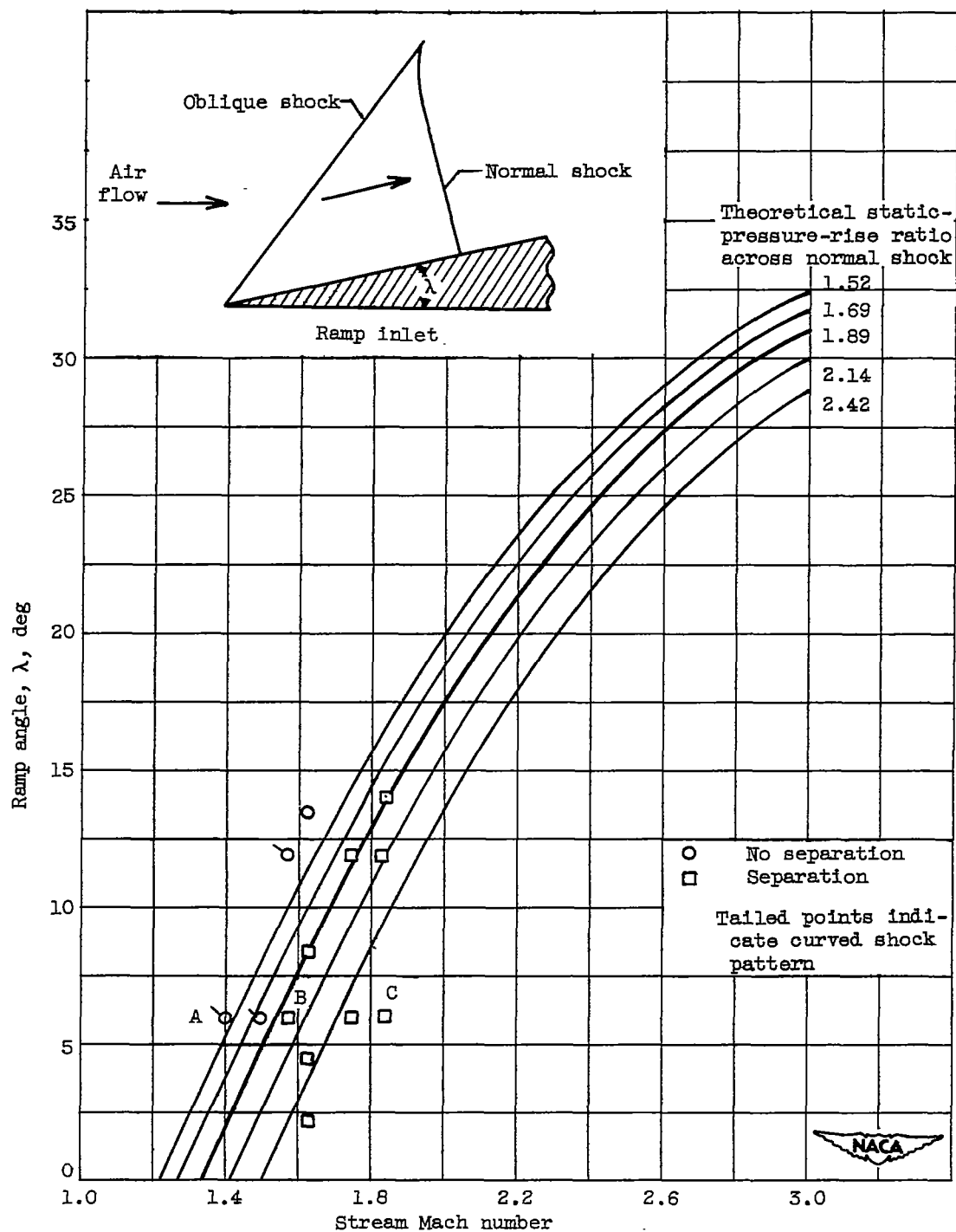


Figure 2. - Relation of ramp angle, Mach number, and theoretical static-pressure-rise ratio across normal shock on two-dimensional inlets. Ratio of specific heats, 1.4.

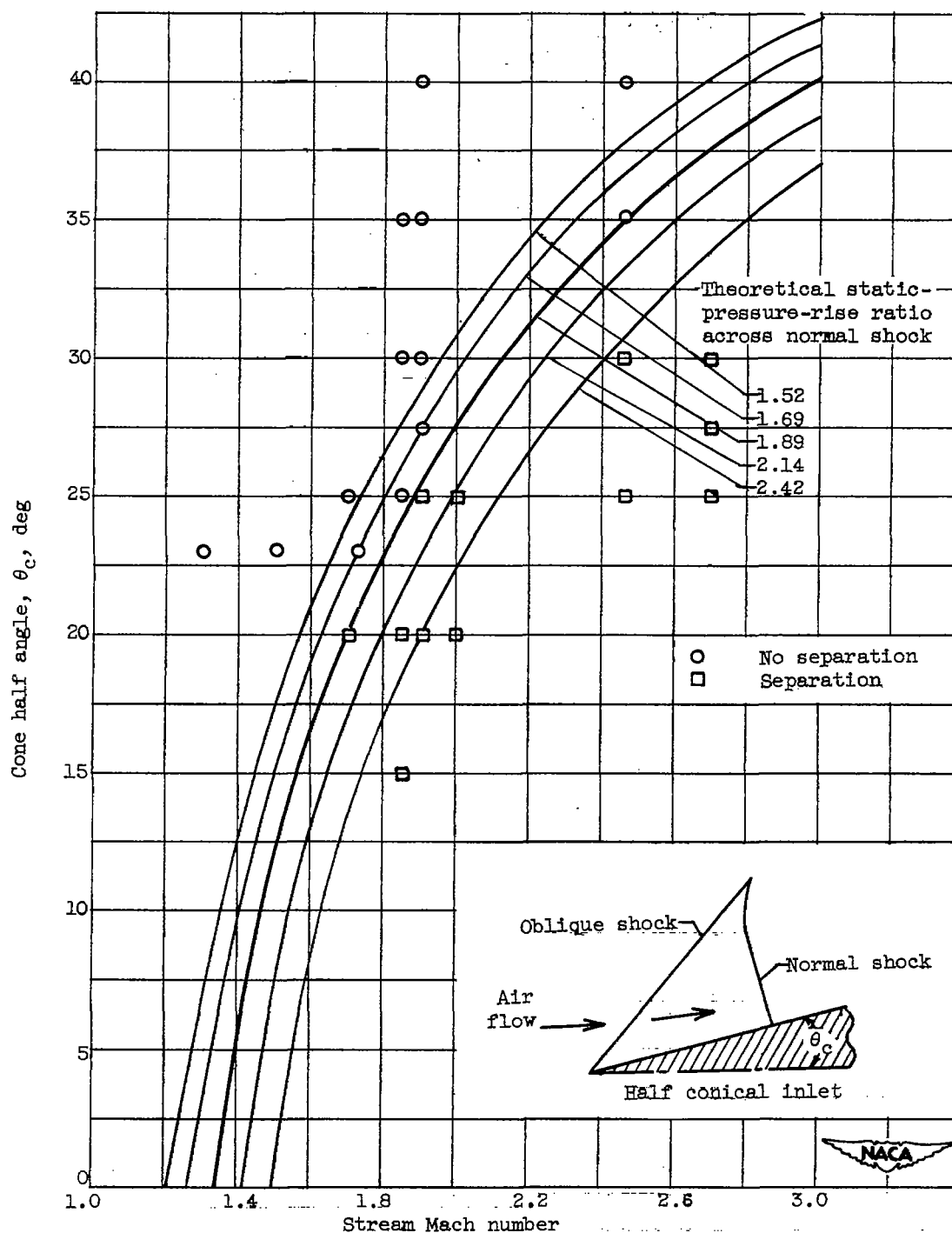


Figure 3. - Relation of cone half angle, Mach number, and theoretical static-pressure-rise ratio across normal shock on cone surface of three-dimensional conical inlets. Ratio of specific heats, 1.4.



(a) Curved shock; Mach number, 1.39; point A of figure 2.



(b) Branched shock; Mach number, 1.57; point B of figure 2.



(c) Branched shock; Mach number, 1.83; point C of figure 2.

Figure 4. - Shock patterns on  $6^\circ$  ramp. Two-dimensional inlet.

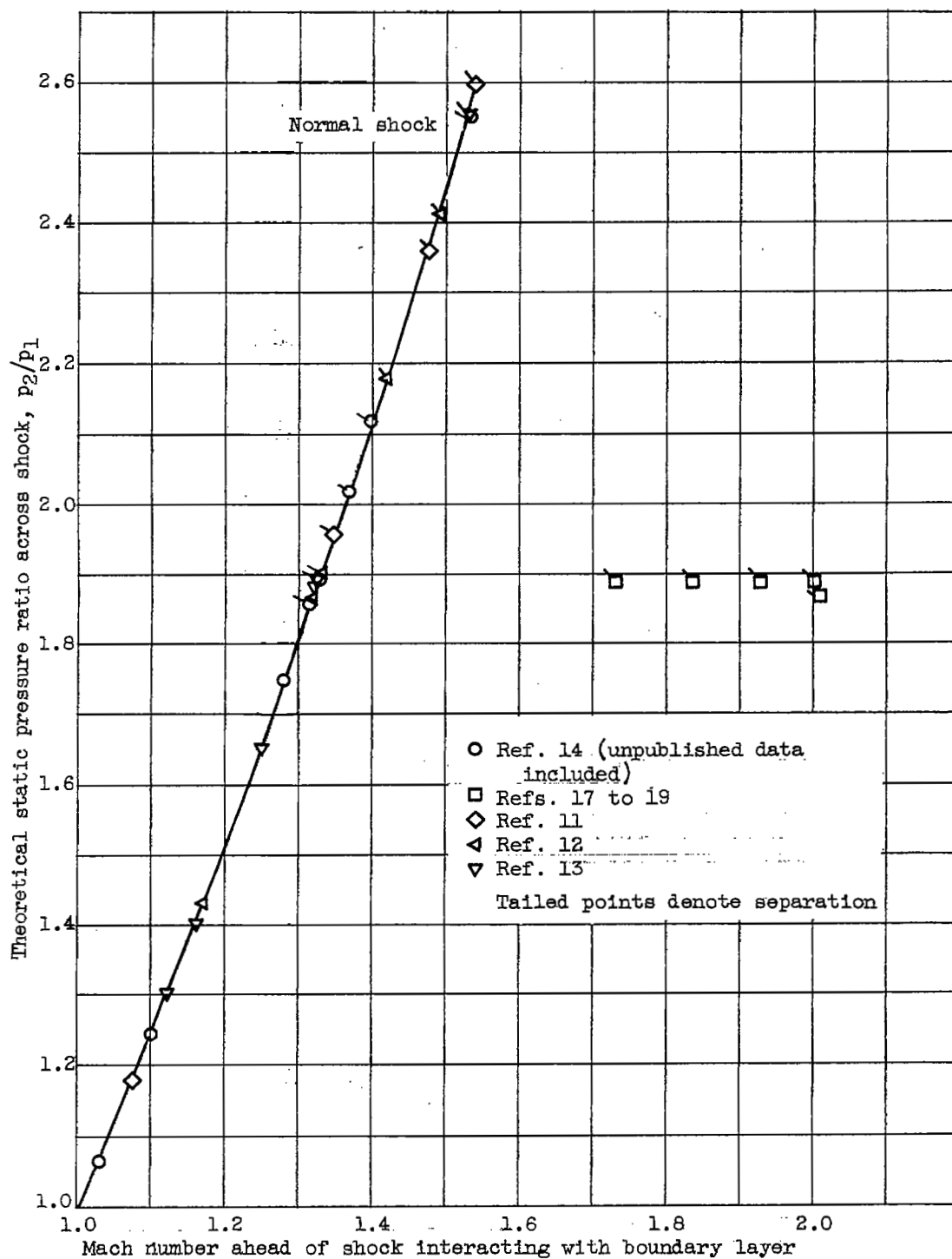


Figure 5. - Correlation of theoretical static pressure rise ratio and Mach number with shock-induced separation; ratio of specific heats, 1.4.

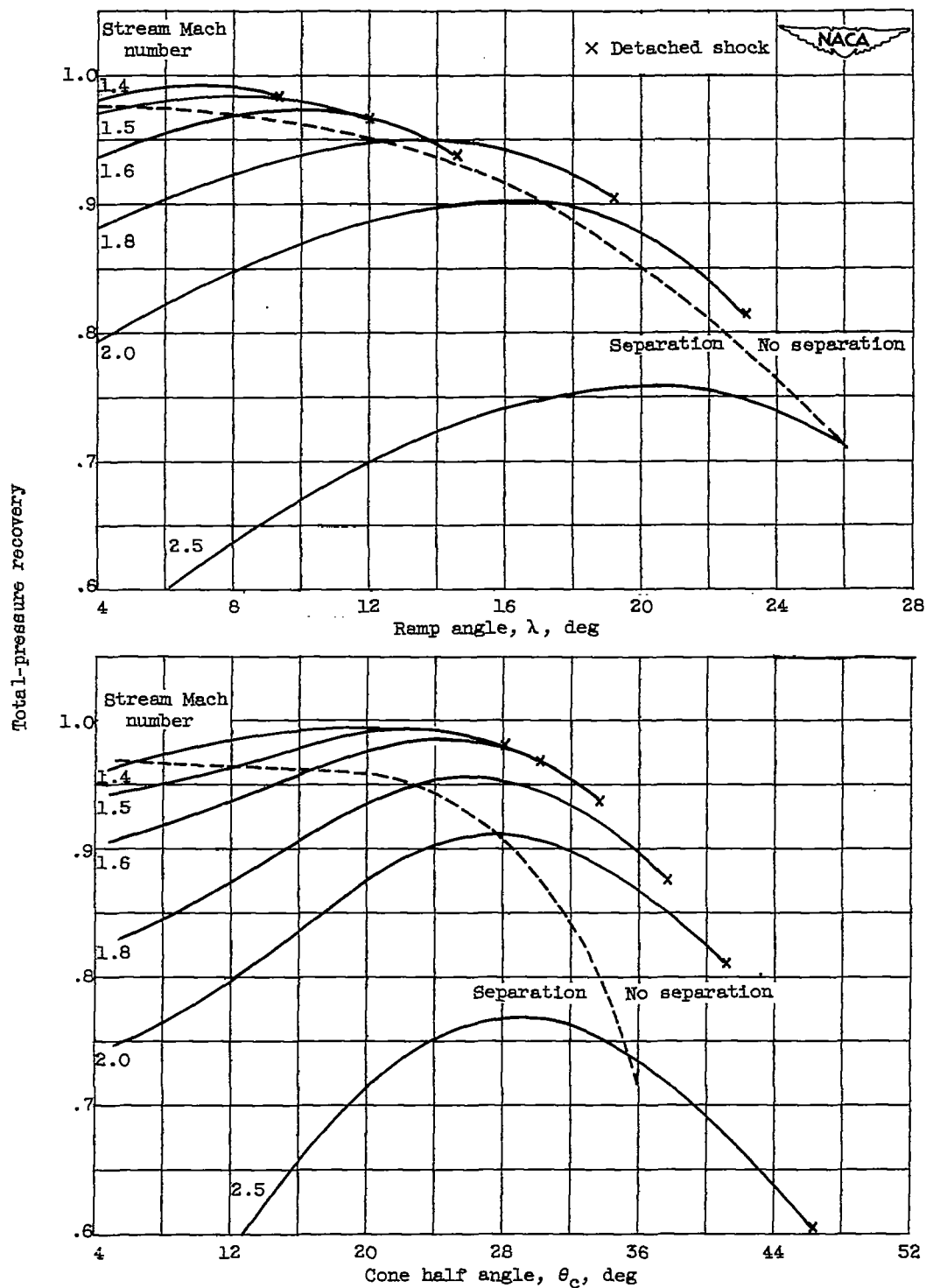


Figure 6. - Theoretical pressure recovery for various two- and three-dimensional inlets. Ratio of specific heats, 1.4.

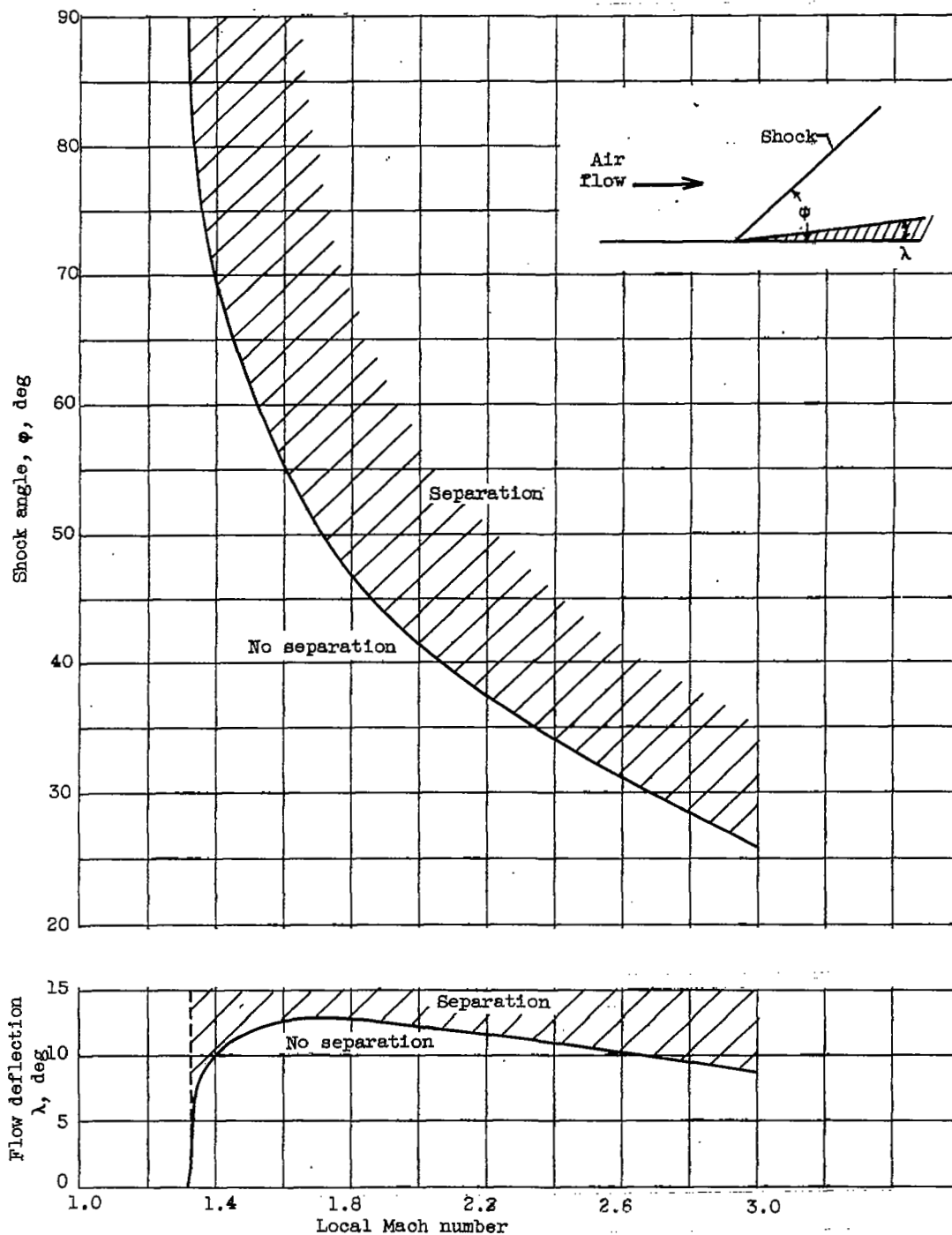


Figure 7. - Effect of Mach number on flow deflection and shock angle in two-dimensional flow required for separation. Ratio of specific heats, 1.4.

